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THE PROBLEM OF AIR-BREATHING JET ENGINES FOR SPACE FLIGHT *[/159]

I.A.Merkulov

Results of a thermodynamic investigation of air-breathing jet engines in the Mach 5 - 50 range are described. Expressions are given for the thrust criterion of an air-breathing jet and it is shown that if heat and an additional mass are delivered to the airflow in this engine it can develop a positive thrust at space flight velocities. Equivalence of heat input is established with respect to engine thrust. It is shown that the mass equivalent of heat decreases with increasing flight velocity. The calculated values of the specific impulse of an air-breathing jet with various types of diffusers at space flight velocities are given.

1. Introduction

The theory of air-breathing jets was developed more than 30 years ago (Bibl.1), and ideas were expressed concerning their use in aviation and for acceleration of space rockets while moving through the atmosphere (Bibl.2). Since then a number of fundamental works (Bibl.3 - 5) have been devoted to varied theoretical and experimental investigations of various types of air-breathing jet engines. The problem of the efficiency of use of air-breathing jets for acceleration of space rockets and the limits of flight velocities up to which such jets can operate have been discussed in recent years in the world's scien-

* Numbers in the margin indicate pagination in the original foreign text.

tific literature (Bibl.6 - 12).

The opinion has been repeatedly expressed in the literature that air-breathing jets can operate only up to comparatively low supersonic velocities corresponding to M = 5 - 6, and that above this limit they cannot be used for various reasons. It has been hypothesized that, apart from the design difficulties involved in developing air-breathing jets for high supersonic velocities, from the theoretical point of view the air-breathing jet generally "degenerates" beyond the indicated velocity limit.

Directly opposite, rather optimistic, opinions have also been expressed concerning the efficiency of air-breathing jets at high, so-called hypersonic velocities.

The purpose of the present paper is to give the general criteria for evaluating the performance of air-breathing jets, i.e., to establish the thrust criteria for these jet engines which would make it possible to elicit, without performing gasdynamic calculations, the capacity of the engine to develop positive thrust at predetermined starting parameters. As the starting parameters we will take: flight velocity, quantities of delivered heat and mass, efficiency of air compression characterized by the efficiency of the diffuser or its pressure coefficient.

As is known, the thrust of an air-breathing jet is expressed by the equation: /160

$$P = 2\left[\mu \frac{V_n}{V_u} - 1\right] S_u \frac{\rho_u V_u^2}{2}, \tag{1.1}$$

where V_u* and V_n are respectively flight velocity and the relative gas discharge

* Here and henceforth the indexes "u", "d", "c", and "n" denote the parameters of the gas-air flow respectively in the undisturbed state in front of the engine, at the end of the diffuser, at the end of the combustion chamber, and at the nozzle exit.

velocity from the motor nozzle; $\mu = 1 + \delta$ is the coefficient of the increase in mass of the gas in the engine; δ is the quantity of fuel or other substances delivered from onboard tanks per kilogram of air passing through the engine; ρ_u is the density of the undisturbed air; S_u is the area of the air stream entering the jet engine.

Since the area and the ram pressure are essentially positive values, the problem of the capability of the engine to develop a positive thrust is wholly determined by the sign of the thrust coefficient

$$C_P = 2\left[\mu \frac{V_n}{V_u} - 1\right].$$

Therefore, as the thrust criterion we can take the inequality

$$\mu \frac{V_n}{V_u} > 1. \qquad (1.2)$$

The operation of the air-breathing jet was investigated under the following assumptions: a) The air and the combustion products are ideal gases conforming to the Clapeyron and Mayer equations; b) the specific heat ratio is constant; c) the gas parameters are constant throughout the cross section; d) gas expansion in the nozzle is complete (i.e., the relationship $p_n = p_u$ exists between the pressures); e) the gas velocities at the end of the diffuser and in the combustion chamber are negligible; f) the gas pressure at the end of the combustion chamber is equal to the pressure at the end of the diffuser ($p_d = p_c$); g) dissociation is absent.

To show more clearly the laws governing the thrust of an air-breathing jet, we will examine in sequence two cases: 1) an air-breathing jet with mechanical energy and heat transfer and no mass transfer (heating of air without delivery of fuel) and 2) a jet with mechanical energy, heat, and mass transfer.

2. Losses of Mechanical Energy in the Air-Breathing Jet Diffuser

As air moves along the duct of the air-breathing jet, irreversible losses of mechanical energy take place, whose magnitude increases with increasing flight velocity. Therefore, when solving the problem of the performance of an air-breathing jet at space flight velocities we must first define the magnitude of these losses.

In the duct of a ramjet type of air-breathing jet engine the greatest portion of the irreversible losses of mechanical energy, as is known, falls to the diffuser. The energy losses as a consequence of friction of the gas as it flows along the combustion chamber walls and nozzle are at least by one order of magnitude less than the losses accompanying the air compression process; thus, special attention should be paid to determining the mechanical energy losses in the diffuser.

The efficiency of air compression (efficiency of the diffuser) can be represented in the form

$$\eta_d = \frac{2g\frac{k}{k-1}RT_u\left[\left(\frac{p_d}{p_u}\right)^{(k-1)/k}-1\right]}{V_u^2}, \tag{2.1}$$

where R is the gas constant, T temperature, and g acceleration of gravity.

Using this equality and bearing in mind the assumption made ($p_n = p_u$, $p_c = p_d$), we will express the thermal efficiency of the Brayton cycle described by the gas in the duct of the air-breathing jet engine in terms of the diffuser efficiency*: /161

$$\eta = \frac{\frac{k-1}{2}M^2\eta_d}{1+\frac{k-1}{2}M^2\eta_d}. \tag{2.2}$$

* For footnote see following page.

The mechanical energy losses during gas expansion in the nozzle can be characterized by the nozzle efficiency:

$$\eta_n = \frac{V_n^2}{2g\frac{k}{k-1}RT_c\left[1-\left(\frac{p_n}{p_c}\right)^{(k-1)/k}\right]} \tag{2.3}$$

With these assumptions, eqs.(2.1) and (2.3) will yield the following expression:

$$\left(\frac{V_n}{V_u}\right)^2 = \frac{T_c}{T_u}\left(\frac{p_u}{p_d}\right)^{(k-1)/k}\eta_d\eta_n. \tag{2.4}$$

To define the effect of the diffuser efficiency on the performance of the air-breathing jet engine and to establish the fundamental laws governing the operation of the engine at space flight velocities, we will determine the losses of kinetic energy in the diffuser. By kinetic energy losses in the diffuser, we mean the difference in the values of the kinetic energy of the airflow entering the diffuser and of the air issuing from the diffuser through an ideal nozzle, with expansion to atmospheric pressure $\Delta E = (V_u^2 - V_x^2) : 2g$. We will refer the value of these losses to the kinetic energy of an undisturbed flow:

$$\Delta \bar{E} = \frac{1-\eta_d}{1-\frac{k-1}{2}M^2\eta_d}. \tag{2.5}$$

In the literature, the diffuser is often characterized by the pressure coefficient, which is the ratio of the air pressure at the end of the diffuser (assuming complete stagnation) to the pressure of an ideally stagnated incident flow

$$\sigma = p_d^* / p_u^*.$$

It is easy to establish the dependence of $\sigma = \sigma(\eta_d)$ and $\eta_d = \eta_d(\sigma)$:

* In determining the efficiency of the Brayton cycle, the pressure drop in the diffuser and the temperature drop of the gas at the end of the diffuser and at the end of the combustion chamber were taken as the starting quantities.

$$\sigma = \left[\frac{1 + \frac{k-1}{2} M^2 \eta_d}{1 + \frac{k-1}{2} M^2} \right]^{k/(k-1)} \tag{2.6}$$

and

$$\eta_d = \frac{\left[1 + \frac{k-1}{2} M^2 \right] \sigma^{(k-1)/k} - 1}{\frac{k-1}{2} M^2}. \tag{2.7}$$

Substituting eq.(2.7) into eq.(2.5), we find /162

$$\Delta \bar{E} = \frac{\left(\frac{1}{\sigma}\right)^{(k-1)/k} - 1}{\frac{k-1}{2} M^2}. \tag{2.8}$$

From eqs.(2.5) and (2.8) we can establish the correlation of

$$\eta_d = (1 - \Delta \bar{E}) \sigma^{(k-1)/k}. \tag{2.9}$$

The qualitative picture of the dependence of the quantities ΔE and $\Delta \bar{E}$ on the Mach number of an incident flow is given by the curves in Fig.1, which were

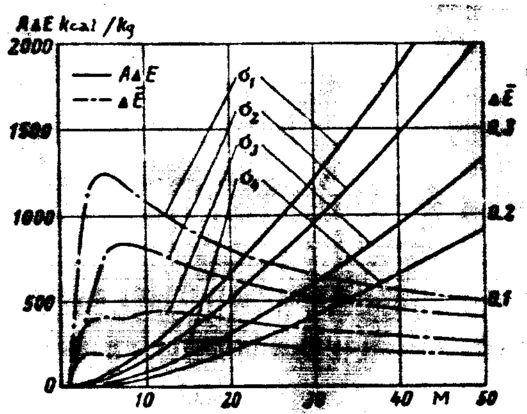

Fig.1

plotted for various values of the pressure coefficient of the duct. (When establishing a correspondence between the quantities M and V_u, the temperature of the ambient air was taken as equal to T_u = 216.66°K, which corresponds to altitudes of 11 - 25 km and 70.7 km.)

As is generally known, no experimental data on the operation of air scoops at space flight velocities are available at present. However, to define correctly the investigation being carried out, we found it advisable to assume certain numerical values of the air-intake pressure coefficients. Therefore, the parameters of the air-breathing jet engine were analyzed for four different scoops. As the lower limit we took a normal-shock intake (σ_1)*. To compare certain concepts of future potentialities of use, we examined three other air-scoops for which, at $M \geq 10$, the following values of the pressure coefficient were taken: $\sigma_2 = 2\sigma_1$, $\sigma_3 = 10\sigma_1$ and $\sigma_4 = 30\sigma_1$.

An analysis of the curves in Fig.1 permits the following conclusions: The absolute value of the kinetic energy losses in the duct of the air-breathing jet engine increases monotonically with an increase of the Mach number of the incident airflow; the relative value of the kinetic energy losses has an extreme, after passing of which the value decreases with increasing Mach number, despite the extremely marked decrease of the pressure coefficient.

3. Engine with Supply of Heat

In order for an air-breathing jet to develop thrust, it is necessary to supply energy to the airflow passing through its duct so as to completely compensate the losses and to ensure that the kinetic energy of the flow in the nozzle exit will exceed the kinetic energy of the airflow entering the engine. The gain in kinetic energy of the airflow as it passes through the air-breathing

* At a normal shock, the pressure drop of the air in the diffuser was arbitrarily assumed as equal to $\pi_d = 166.923\ M^2/[7 - 1/M^2]^{25}$. Allowance for the change in the specific heat of the air, with respect to the estimate made by the author for $M = 5 - 30$, increases the value of the drop by not more than 5%.

jet is equivalent to the development of thrust in the case of equality of the entering and discharging air masses. When the mass of the gas increases as it moves along the duct of the engine, as will be shown below, the generation of thrust is possible at a constant, and even at a reduced, value of the kinetic energy of the gas flow.

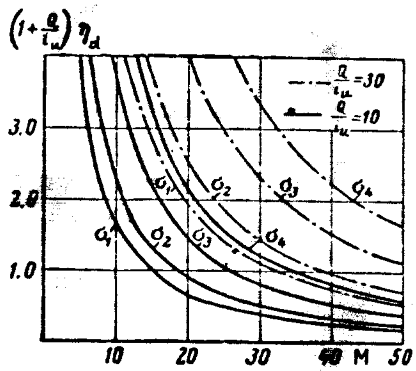

Fig.2

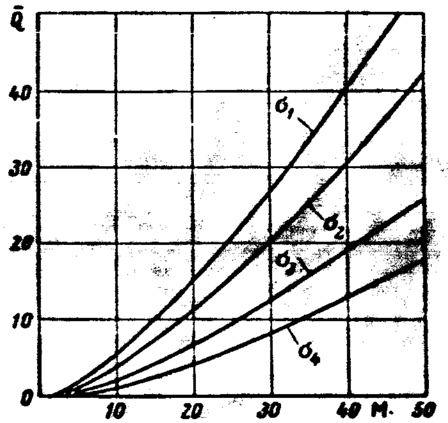

Fig.3

If heat is supplied to the airflow passing through the engine while /163
maintaining its mass constant, eqs.(2.4) and (2.1) will furnish

$$\left(\frac{V_u}{V_n}\right)^2=\frac{T_u}{T_c}\left[\frac{k-1}{2}M^2+\frac{1}{\eta_d}\right]\frac{1}{\eta_n}. \qquad (3.1)$$

The criterion of thrust development is represented by the expression

$$\frac{T_c}{T_u}>\left[\frac{k-1}{2}M^2+\frac{1}{\eta_d}\right]\frac{1}{\eta_n}, \qquad (3.2)$$

which shows that, at large values of the Mach number, the influence of the quantity η_d on the required ratio of heated-gas temperature to atmospheric air diminishes.

From the inequality (3.2) we easily find the criterial dependence, expressed in terms of the quantity of heat delivered per kilogram of air.

Since $T_c/T_u = 1 + (k - 1/2)M^2 + Q/c_p T_u$, we have

$$\left[1+\frac{Q}{i_u}-\frac{1-\eta_n}{\eta_n}\frac{k-1}{2}M^2\right]\eta_d\eta_n>1. \qquad (3.3)$$

Setting $\eta_n = 1$, 0 we obtain

$$\left(1+\frac{Q}{i_u}\right)\eta_d>1. \qquad (3.4)$$

This expression represents the thrust criterion of a ramjet, expressed in terms of three basic quantities: enthalpy of the air characterizing the external environment; quantity of energy delivered to the air in the combustion chamber, and diffuser efficiency. Figure 2 shows the dependence of the left-hand side of the inequality (3.4) on the Mach number, for various types of diffusers and for two values of the quantity Q (i_u = 51.74 kcal/kg).

For convenience of practical application of the obtained criterion to various calculations, we will write it in several forms.

To determine the value of Q at which the engine thrust P > 0, the thrust criterion can be presented in the form

$$\bar{Q}=\frac{Q}{i_u}>\frac{1-\eta_d}{\eta_d}. \qquad (3.5)$$

Figure 3 shows the dependence of the criterion $\bar{Q}$ on the Mach number for /164
several types of airscoops.

The curves in Fig.3 permit establishing the Mach number at which the air-breathing jet, working with a predetermined delivery of heat (Q) and having a specified air intake (σ), ceases to develop a positive thrust. We should point out that the limit of practical use of the engine is located at appreciably lower Mach numbers than the limit of zero thrust.

It is also possible to state the problem in a different form: Let us determine the lower limit of the values of η_d which satisfy the condition P > 0 at a given value of Q. In this case, the thrust criterion should be written as

$$[illegible] \qquad (3.6)$$

If the value of σ is given but not that of η_d, then the critical expressions can be transformed, on the basis of eqs.(3.4) and (2.7), in the following manner:

$$\frac{\left(1+\frac{k-1}{2}M^2\right)\left(1+\frac{Q}{i_u}\right)}{1+\frac{k-1}{2}M^2+\frac{Q}{i_u}}\sigma^{(k-1)/k} > 1. \qquad (3.7)$$

The thrust criterion of an air-breathing jet can be represented in two other forms if we express it in terms of absolute flight velocity or enthalpy of the airflow*:

$$\frac{\left(i_u + A\frac{V_u^2}{2g}\right)(Q+i_u)}{\left(i_u + A\frac{V_u^2}{2g}+Q\right)i_u}\sigma^{(k-1)/k} > 1 \qquad (3.8)$$

and

$$\frac{i_u^*(i_u+Q)}{i_u(i_u^*+Q)}\sigma^{(k-1)/k} > 1. \qquad (3.9)$$

Analogously, from the inequalities (3.5) and (3.6) we can obtain

$$\bar{Q} = \frac{Q}{i_u} > \frac{\left(1+\frac{k-1}{2}M^2\right)\left(1-\sigma^{(k-1)/k}\right)}{\left(1+\frac{k-1}{2}M^2\right)\sigma^{(k-1)/k}-1}, \qquad (3.10)$$

$$\bar{Q} > \frac{\left(i_u + A\frac{V_u^2}{2g}\right)\left(1-\sigma^{(k-1)/k}\right)}{\left(i_u + A\frac{V_u^2}{2g}\right)\sigma^{(k-1)/k}-1} \qquad (3.11)$$

$$\bar{Q} > \frac{i_u^*\left(1-\sigma^{(k-1)/k}\right)}{i_u^*\sigma^{(k-1)/k}-i} \qquad (3.12)$$

and

$$\sigma > \left[\frac{\left(1+\frac{k-1}{2}M^2\right)i_u+Q}{\left(1+\frac{k-1}{2}M^2\right)(i_u+Q)}\right]^{\cdot} \qquad (3.13)$$

---

* The asterisk denotes parameters of an isentropically stagnated flow.

The resultant expressions for the thrust criterion are valid not only for ramjets but also for air-breathing jets of other types (for example, turbo-ramjets and turbojets) provided that we extend the concept of the pressure coefficient, understanding it as the ratio of the gas pressure ahead of the nozzle to the pressure of an ideally stagnated airflow ahead of the engine. /165

4. Engine with Supply of Heat and Mass

Let us find the expression of the thrust criterion, for the case where some mass is supplied to the airflow passing through the engine. As the mass being supplied we can use inert substances, fuel, oxidizer, and their various combinations.

In this case, the thrust criterion takes the form

$$\frac{i_c}{i_u} > \frac{1}{\mu^2}\left(\frac{k-1}{2}M^2 + \frac{1}{\eta_d}\right)\frac{1}{\eta_n} \qquad (4.1)$$

Since $\mu i_c = i_d + \delta i_T + Q$, then, having designated $Q_\Sigma = Q + \delta i_T$ (i_T being the enthalpy of the supplied mass), we obtain:

$$\frac{i_c}{i_u} = \frac{i_d + Q_\Sigma}{\mu i_u} = \frac{1}{\mu}\left(1 + \frac{k-1}{2}M^2 + \frac{Q_\Sigma}{i_u}\right).$$

Consequently,

$$\left(1 + \frac{k-1}{2}M^2 + \frac{Q_\Sigma}{i_u}\right)\frac{1}{\mu} > \frac{1}{\mu^2}\left(\frac{k-1}{2}M^2 + \frac{1}{\eta_d}\right)\frac{1}{\eta_n}.$$

Hence

$$\left[\left(1 + \frac{Q_\Sigma}{i_u}\right) + \left(1 - \frac{1}{\mu\eta_n}\right)\frac{k-1}{2}M^2\right]\mu\eta_d > 1, \qquad (4.2)$$

or, setting η_n = 1, 0,

$$\left[1 + \frac{Q_\Sigma}{i_u} + \frac{\delta}{1+\delta}\,\frac{k-1}{2}M^2\right](1+\delta)\eta_d > 1. \qquad (4.3)$$

Using expression (2.7) we can obtain the thrust criteria expressed in terms of σ

$$\frac{\left[1+\frac{k-1}{2}M^2\right]\left[1+\frac{Q_\Sigma}{i_u}+\frac{\delta}{1+\delta}\,\frac{k-1}{2}M^2\right]}{1+\frac{k-1}{2}M^2+\frac{Q_\Sigma}{i_u}}\sigma^{(k-1)/k}>1, \qquad (4.4)$$

$$\frac{\left[i_u+A\frac{V^2}{2g}\right]\left[i_u+Q_\Sigma+\frac{\delta}{1+\delta}\,\frac{AV^2}{2g}\right]}{\left[i_u+A\frac{V^2}{2g}+Q_\Sigma\right]i_u}\sigma^{(k-1)/k}>1 \qquad (4.5)$$

or

$$\frac{i_u^*\left[i_u+Q_\Sigma+\frac{\delta}{1+\delta}(i_u^*-i_u)\right]}{i_u[i_u^*+Q_\Sigma]}\sigma^{(k-1)/k}>1. \qquad (4.6)$$

An analysis of the thrust criterion of an air-breathing jet shows that the greater the flight velocity, the greater will be the effect from the addition of an auxiliary mass to the jet engine.

Let us examine the case $Q_\Sigma = 0$. From the inequality (4.3), we can obtain /166

$$\delta>\frac{1-\eta_d}{\left(1+\frac{k-1}{2}M^2\right)\eta_d}, \qquad (4.7)$$

which is the condition of thrust development without supply of heat.

Remembering eq.(2.7), the inequality (4.7) will also yield the following expression:

$$\left[1+\frac{\delta}{1+\delta}\,\frac{k-1}{2}M^2\right]\sigma^{(k-1)/k}>1. \qquad (4.8)$$

Hence, using eq.(2.8) we can also establish

$$\frac{\delta}{1+\delta}>\frac{\left(\frac{1}{\sigma}\right)^{(k-1)/k}-1}{\frac{k-1}{2}M^2}\equiv\Delta\bar{E} \qquad (4.9)$$

and

$$\delta>\frac{\Delta\bar{E}}{1-\Delta\bar{E}}. \qquad (4.10)$$

Equation (4.10) shows the quantity of mass that must be added to the duct of a "cold" air-breathing jet ($Q_\Sigma = 0$) to compensate the kinetic energy losses (Fig.4). Any noticeable mechanical energy losses during mixing of the additional mass with the air can be taken into account by the values of η_d or σ.

The noted laws permit formulating the following theorem on the equivalence of additional masses and additional energy from the point of view of thrust production.

"The mass delivered to an airflow passing through an air-breathing jet engine is equivalent to the added heat, and when the relative values of added mass and heat are equal the engine will develop a uniform thrust."

Actually, if we first assume in eq.(4.3) $Q_\Sigma = 0$ and then $\delta = 0$ and if we then equate both values of the criterion, bearing in mind that the equality of the quantities of the criterion indicates equality of the thrust magnitudes, we obtain

$$\delta = \frac{Q_\Sigma / i_u}{1 + \frac{k-1}{2} M^2} \quad \text{or} \quad \delta = \frac{Q_\Sigma}{i_u^*}.$$

With an increase in flight velocity, the magnitude of the equivalent additional mass, i.e., the mass ensuring the development of the same thrust as at supply of unit heat, decreases continually.

The equivalence coefficient is a monotonically decreasing function of the Mach number

$$\bar{\delta} = \frac{\delta}{Q_\Sigma / i_u} = \frac{1}{1 + \frac{k-1}{2} M^2}.$$

For example, for M = 4 a heat input amounting to one large calorie per kilogram of air is equivalent to the delivery of about 5×10^{-3} kilogram of an inert mass per kilogram of air. For M = 10, the value of the equivalent mass drops approximately to 10^{-3} and, for M = 30, to 10^{-4}.

The thrust criterion of an air-breathing jet engine expressed by eq.(4.3) permits determining the minimal value of the heat input at which the engine /167 begins to develop a positive thrust at a given value of the mass flow rate δ. Toward this end, let us solve eqs.(4.3 and (4.4) relative to the quantity $\overline{Q}_\Sigma$

$$\overline{Q}_\Sigma = \frac{Q_\Sigma}{i_u} > \frac{1}{1+\delta}\left[\frac{1-\eta_d}{\eta_d} - \delta\left(1+\frac{k-1}{2}\mathrm{M}^2\right)\right]$$

and

$$\overline{Q}_\Sigma = \frac{\left[1+\frac{k-1}{2}\mathrm{M}^2\right]\left[1-\sigma^{(k-1)/k}\left(1 \text{[illegible]} \frac{k-1}{2} \text{[illegible]}\right)\right]}{\left[1+\frac{k-1}{2}\mathrm{M}^2\right]\sigma^{(k-1)/k}-1}.$$

Figure 5 shows the results of calculating the limiting value of $\overline{Q}_{\Sigma_{lim}}$ for various flow rates (for σ_1).

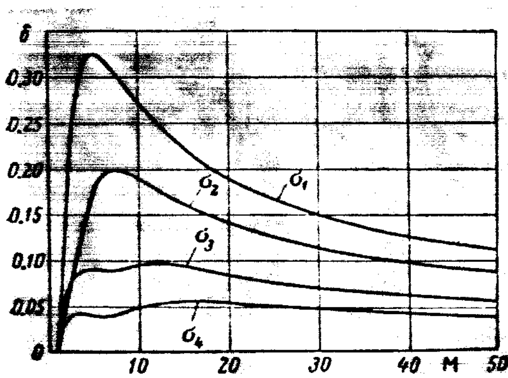

Fig.4

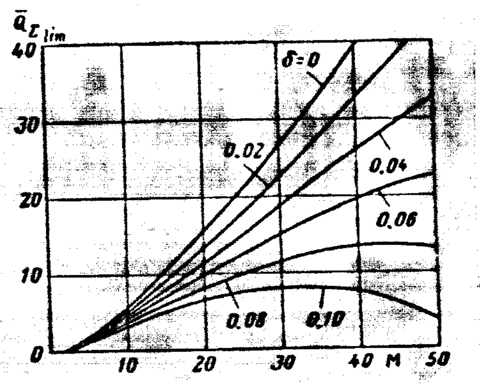

Fig.5

Analogous to eqs.(3.6) and (3.13), we can find the limiting values of the quantities η_d or σ at which the condition $P > 0$ is satisfied for given values of $\overline{Q}_\Sigma$ and M. From eq.(4.3) we find

$$\eta_d > \frac{1}{\frac{i_u+Q_\Sigma}{i_u} + \delta\left[\frac{i_u+Q_\Sigma}{i_u} + \frac{k-1}{2}\mathrm{M}^2\right]} \tag{4.11}$$

and, from eq.(4.4)

$$\sigma > \left[\frac{1+\frac{k-1}{2}\mathrm{M}^2+\frac{Q_\Sigma}{i_u}}{\left(1+\frac{k-1}{2}\mathrm{M}^2\right)\left(1+\frac{Q_\Sigma}{i_u}+\frac{\delta}{i+\delta}\frac{k-1}{2}\mathrm{M}^2\right)}\right]^{k/(k-1)} \tag{4.12}$$

As shown in Fig.6, at $\delta = 0$ for $M \geq 21$ the required value of the coefficient σ_{lim} exceeds the value of σ for a normal shock and, unless diffusers with a higher pressure coefficient are developed, the air-breathing jet will not be able to develop positive thrust. At $\delta > 0$, the air-breathing jet can work with an air intake pressure coefficient even lower than in the case of a normal shock.

5. Specific Impulse of Space Flight Air-Breathing Jet Engines

A logical question is that concerning the increase in mass rate of flow. The solution of this problem is affected by the following three factors: required value of absolute engine thrust, limiting value of permissible gas /168
temperature in the combustion chamber, and dependence of the specific impulse on the mass rate of flow.

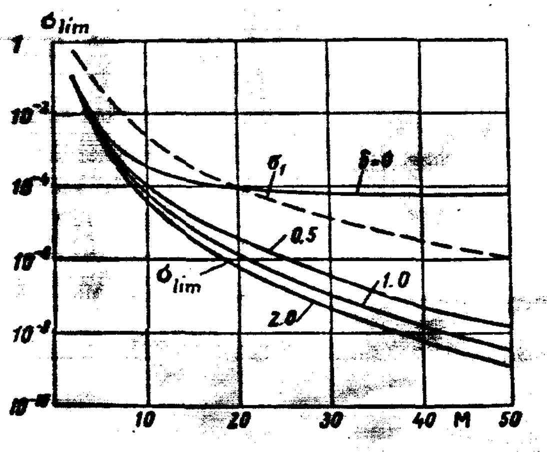

Fig.6

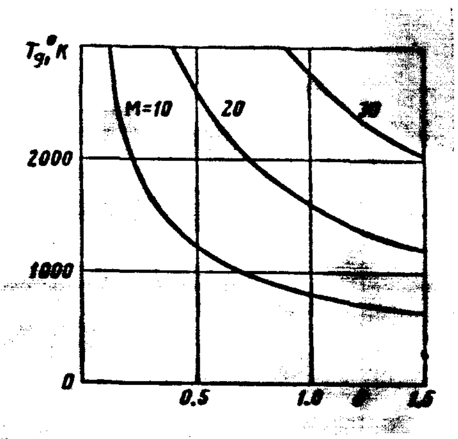

Fig.7

The inequalities (4.3) - (4.6) show that the thrust of an air-breathing jet increases without limit with an increase in mass flow rate. The gas temperature T_g monotonically decreases with an increase in the amount of mass supplied to the airflow (Fig.7). Thus, the first two of the three above factors improve with an increase in mass flow rate. The manner in which the mass flow rate affects the specific impulse must be determined next. At what mass flow rate does the specific impulse reach a maximal value?

The value of the specific impulse can be calculated from the formula

$$I=\left(\frac{\mu V_n}{V_u}-1\right)\frac{V_u}{g\delta}. \qquad (5.1)$$

Expressing the gas discharge velocity in the form of

$$V_n=\sqrt{2g\frac{i_u^*+Q_\Sigma}{A\mu}\,\frac{\frac{k-1}{2}M^2\eta_d}{1+\frac{k-1}{2}M^2\eta_d}}$$

we obtain

$$I=\left[\sqrt{(1+\delta)\frac{1+\frac{k-1}{2}M^2+\bar{Q}_\Sigma}{1+\frac{k-1}{2}M^2\eta_d}\eta_d}-1\right]\frac{Ma_u}{g\delta}, \qquad (5.2)$$

where a_u is the speed of sound in an undisturbed flow.

To determine the extreme value of the function $I = I(\delta)$, we will analyze its first derivative

$$\frac{dI}{d\delta}=\frac{Ma_u}{d\delta^2}\left[1-\frac{2+\delta}{2\sqrt{1+\delta}}\sqrt{\frac{1+\frac{k-1}{2}M^2+\bar{Q}_\Sigma}{1+\frac{k-1}{2}M^2\eta_d}\eta_d}\right]=0 \qquad (5.3)$$

and obtain /169

$$\delta_{extr.}=2\frac{1-\frac{1+\frac{k-1}{2}M^2+\bar{Q}_\Sigma}{1+\frac{k-1}{2}M^2\eta_d}\eta_d+\sqrt{1-\frac{1+\frac{k-1}{2}M^2+\bar{Q}_\Sigma}{1+\frac{k-1}{2}M^2\eta_d}\eta_d}}{\frac{1+\frac{k-1}{2}M^2+\bar{Q}_\Sigma}{1+\frac{k-1}{2}M^2\eta_d}\eta_d}. \qquad (5.4)$$

Investigating the trend of the function $\delta_{extr} = \delta_{extr}(M)$ we can establish that the supply of an additional mass will lead to an increase in specific impulse only at certain values of the Mach number corresponding to the condition:

$$\frac{1+\frac{k-1}{2}M^2+\bar{Q}_\Sigma}{1+\frac{k-1}{2}M^2\eta_d}\eta_d < 1. \tag{5.5}$$

This inequality permits finding the velocity above which it is suggested to supply an additional mass for increasing the specific impulse. The sought

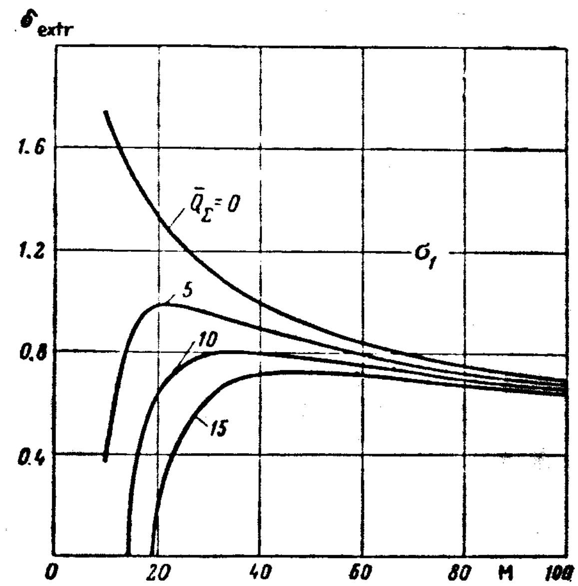

Fig.8

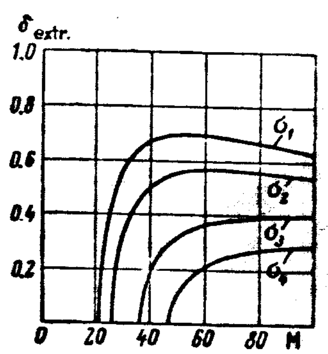

Fig.9

velocity value depends on the relative value of the heat input $\bar{Q}$ and the diffuser efficiency. At lower flight velocity, the supply of an additional mass can be justified by the desire to increase the absolute thrust of the engine, by the need to lower the gas temperature in the combustion chamber, and by problems of cooling the engine components. The additional mass can be supplied for injection pressurizing of the engine and for other purposes.

At an infinite increase in Mach number, lim $\delta_{extr} = 0$.

Since the function $\delta_{extr} = \delta_{extr}(M)$ twice becomes equal to zero, we can establish that it has a maximum and can thus determine its value as well as the value of the Mach number at which this maximum is realized.

For example, for $\overline{Q} = 16.4$ and $\sigma = \sigma_1$, we find $\delta_{extr} = 0.7$ kg per kilogram of air and $M = 50$. /170

Consequently, in the entire range of $0 < M < \infty$, for values of $\overline{Q} = 16.4$ and $\sigma = \sigma_1$, the maximal quantity of additional mass supplied to increase the specific impulse does not exceed 0.7 kg/kg of air.

The exception to this rule is the case in which the increase in the flow rate of the additional mass causes a change in the quantity $\overline{Q}_{\Sigma}$ or an increase in the gas pressure in the combustion chamber (injection pressurizing, operation of an air-breathing jet engine with a fixed-area nozzle with an oversize critical throat section).

Figures 8 and 9 show the dependence on the Mach number and $\overline{Q}_{\Sigma}$ of the value of δ_{extr}, realizing the maximum of the function $I(\delta)$ for various types of diffusers. The slope of the curves in these diagrams indicates that, with an increase in the pressure coefficient of the diffuser, the maximum of the function $\delta_{extr}(M)$ shifts to larger M values so that, in this case, the value of $(\delta_{extr})_{max}$ decreases.

Figure 10 illustrates the dependence of the specific impulse on the /171 amount of substances supplied per kilogram of air in the combustion chamber.

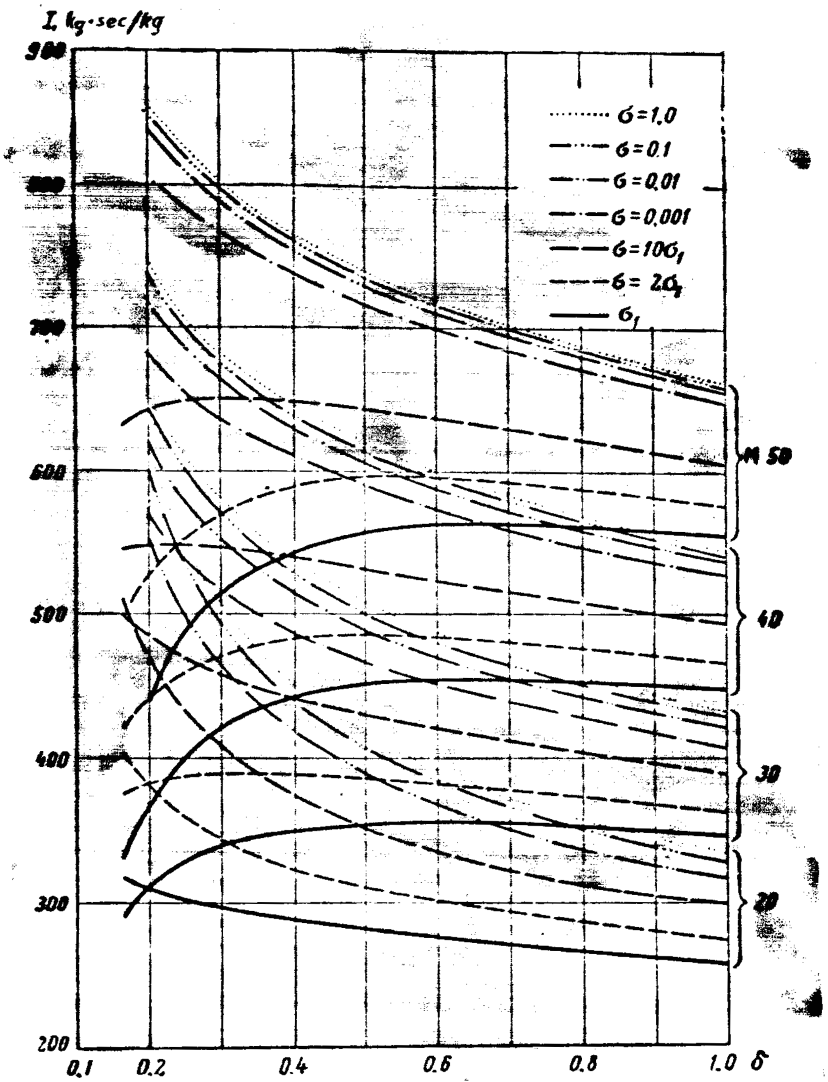

Fig.10

Thus, the results of a calculation performed by an approximation method, with the above assumptions, showed that the investigated engine with a normal-shock diffuser at flight velocities corresponding to M = 30 - 40 has a specific impulse of the same order as modern liquid-fuel rocket engines, differing from them by larger size and weight; however, in the case of developing a diffuser with an appreciably greater pressure coefficient than the normal-shock diffuser, the engine in this velocity range may have an impulse on the order of

500 - 600 kg • sec/kg.

With an increase in flight velocity, the value of the specific impulse of the investigated engine substantially increases and, at $M = 50$, approaches 600 kg • sec/kg.

These values of the specific impulse appreciably exceed the values which current notions assign to prospective liquid-fuel rocket engines.

This conclusion indicates the advisability of investigating and perfecting airscoops intended for use at flight velocities of the order of the first-to-third cosmic velocities (orbital velocity, escape velocity, and escape velocity from the solar system).

The high value of the specific impulse of the investigated engine at flight velocities exceeding the escape velocity (including those with a normal-shock air intake) makes it expedient to examine the possibility of using such engines in the upper atmosphere, for accelerating to a velocity of $V = 15 - 18$ km/sec any spacecraft intended for orbits of far-out planets or vehicles that require such velocities for other purposes (to reduce flyby time).

6. Conclusions

1. An air-breathing jet engine in which heat and an additional mass is supplied to the flow is able to develop a positive thrust at space-flight velocities.

2. Irreversible kinetic energy losses in the diffuser of the air-breathing jet monotonically increase with increasing flight velocity and, in the case of a normal shock at orbital velocity, reach an order of magnitude of 4×10^6 Joule/kg of air. However, the magnitude of the losses with respect to the total kinetic energy of the airflow entering the engine, having reached a maximum

(25%) at a Mach number of M = 5, decreases with further increase in velocity and, at orbital velocity, amounts to only 13% for a normal-shock diffuser.

3. The supply of mass is equivalent to the supply of heat with respect to the value of engine thrust produced. The mass equivalent of heat decreases with an increase in flight velocity and, for example, at M = 30 becomes equal to 10^{-4} kg/kcal.

4. The supply of an inert mass at Mach numbers exceeding 15 - 20 will lead both to an increase in absolute thrust and an increase in specific impulse (for a normal-shock engine and at $\overline{Q}_{\Sigma} = 10 - 15$).

5. The optimal value of the quantity of mass supplied to the airflow corresponding to the maximal specific impulse varies with the flight velocity. For example, at $\overline{Q}_{\Sigma} = 15$ for an engine having a normal-shock diffuser, we have $\delta_{extr} \approx 0.6$ at M = 30, and $\delta_{extr} \approx 0.7$ at M = 40.

6. The quantity δ_{extr} has a maximum, whose value and the velocity at which it is reached depend on the type of diffuser and on the amount of heat supplied (regardless of the amount of inert mass); for example, at $\overline{Q}_{\Sigma} = 10$ the maximal value will be $\delta_{extr} = 0.8$ (M = 35), and,at $\overline{Q}_{\Sigma} = 15$, $\delta_{extr} = 0.73$ (M = 50).

7. In the range of orbital to escape velocity (M ≈ 30 - 40) the air-breathing engine with a normal-shock diffuser has a specific impulse of the same order as that of modern liquid-fuel rocket engines; for a diffuser whose /172 pressure coefficient is ten times higher than that of a normal-shock diffuser (at M = 40 $\sigma_3 \approx 0.00004$), the value of the specific impulse increases to 500 - 550 kg • sec/kg. At a flight velocity corresponding to M = 60, the specific impulse of an engine with the indicated diffusers reaches 700 - 750 kg • • sec/kg.

8. All of the conclusions are valid for altitudes at which the airflow

entering the engine can be considered a continuum. The gas flow in the combustion chamber is taken as subsonic. The engine thrust was investigated while neglecting the external drag of the engine nacelles.

The numerical values of certain quantities given in this paper are approximate, since they were obtained on the assumption of constancy of the heat capacity of the gas, complete expansion of the gas in the nozzle, absence of losses during mixing of the gases in the combustion chamber, absence of energy losses for cooling, and absence of energy losses due to dissociation.

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